Current MBDA R&T Effort on Ram/Scramjet and Detonation Wave Engine

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ABSTRACT

Since the fifties, MBDA France, in close cooperation with ONERA, has been leading a sustained effort to develop ramjet technology. This strong involvment led to the development of the Air Sol Moyenne Portée (ASMP or Medium Range Air-to-Ground) missile which entered service in the French Air Force in 1986 and will be soon replaced by ASMP A.

Beyond this development, a lot of advanced studies were performed during the nineties to improve the technology, to increase the accessible flight envelope and to develop needed numerical and experimental means. Increasing the range was one of the main objectives. Many studies of new technologies for thermal insulation, permitting to drastically extend the operating time, were realized. In parallel new airbreathing missile configurations were studied for decreasing the natural Radar Cross Section or increasing cruise speed and altitude. Complementary studies were performed to improve the volumic impulse of liquid fuels particularly by using Boron slurries.

Today, the R&T effort on ramjet technology is focused on two main axis :

- *extension of the ramjet technology to higher Mach numbers (up to Mach 8 for military application and up to Mach 12 for space launcher application) by developing needed technologies for dual-mode ramjet and flight testing of the experimental vehicle LEA,*
- *reduction of the development and production costs by developing the modular low cost RASCAL concept of liquid hydrocarbon fuel ramjet for tactical missiles.*

Due to its thermodynamic cycle, a detonation wave engine has theoretically a higher performance than an other classical propulsion concept using the iso-pressure combustion process. Nevertheless, it still has to be proven that this advantage is not compensated by the difficulties which could be encountered to practically design a real engine and to implement it into an operational flying system.

During past years, MBDA France performed some theoretical and experimental works on Pulsed Detonation Engine (PDE), mainly in cooperation with LCD laboratory at ENSMA Poitiers and CIAM and Semenov Institutes in Moscow, in order to obtain a preliminary demonstration of the feasibility of the PDE in both rocket and airbreathing modes and to verify the interest of such a PDE for operational application. On this basis, several engine concepts have been studied and evaluated at preliminary design level, for both space launcher and missile application. Today, the effort is focused on the development of a small caliber airbreathing engine able to power a UAV with very demanding requirements in terms of Thrust range. Such an engine could be flight tested within the next years.

The use of a Continuous Detonation Wave Engine (CDWE) can also be considered to reduce the environmental conditions generated by PDE while reducing the importance of initiation issue and

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simplifying some integration aspects. MBDA France is leading a specific R&T program, including basic studies with the Lavrentiev Institute of Novosibirsk, to assess some key points for the feasibility of an operational rocket CDWE for space launcher.

But, continuous detonation wave can have also other application for turbojets and for ramjets. In order to address all these possible applications, a ground demonstrator has been designed and should be developed and tested within the next years.

1. INTRODUCTION

As the propulsion system is a key sub-system for missiles, MBDA France (and its former components) has always been leading a lot of effort to develop the related technologies (generally with specialized partners) and to master their optimum integration into its missile products. This approach is particularly developed for the ramjet technology since the fifties but, today, a new field is also explored with a renewed interest for the detonation wave engines.

1.1 Ramjet/scramjet technology

The strong R&D effort led by MBDA France for decades obtained a first consecration with the development of the Air Sol Moyenne Portée (ASMP or Medium Range Air-to-Ground) missile which entered service in the French Air Force in 1986 and will be soon replaced by ASMP A. By another way, the Anti Navire Supersonique (ANS or Supersonic Anti-Ship) missile predevelopment culminated, in the late 1980’s, in three successful flight tests, further expanding and demonstrating MBDA France know-how in high performance ramjets and particularly its ability to perform very low altitude supersonic sea-skimming.

Beyond these developments a lot of advanced studies were performed during the nineties to improve the technology, to increase the accessible flight envelope and to develop needed numerical and experimental means. Increasing the range was one of the main objectives. Many studies of new technologies for thermal insulation, permitting to drastically extend the operating time, were realized. Thermal insulations constituted by a ceramic composite heat shield protecting from erosion and by a layer of insulation material were tested and solutions, able to sustain low altitude terminal penetration phase, were finally qualified. Film cooling by air injection was also successfully developed but the compatibility with integral booster is still to be fully demonstrated. In parallel new airbreathing missile configurations were studied for decreasing the natural Radar Cross Section (ASLP concept using a single inlet placed on top of the fuselage) or increasing cruise speed and altitude (MARS concept). Complementary studies were performed to improve the volumic impulse of liquid fuels. Boron slurries, containing up to 70 % of micronic and submicronic particles of boron were tested in a ramjet combustion chamber with good performances while the feasibility of its use on board of an operational missile was studied thanks to a specific Advanced development program.

All these works were conducted with the permanent use and development of numeric tools. Many efforts have been done in that way to increase prediction before test and analysis after test. 3D computations are now daily used for designing vehicle aerodynamic configuration and air inlets. Numerical simulation of combustion is commonly used to compare ramjet configurations before test or for analyzing flame stabilization. However, combustion instabilities or transition phases are not yet predictable and still need many tests.

Today, the R&T effort on ramjet technology is focused on two main axis :

- extension of the ramjet technology to higher Mach numbers (up to Mach 8 for military application and up to Mach 12 for space launcher application) by developing needed technologies for dual-mode ramjet and flight testing of the experimental vehicle LEA,
- reduction of the development and production costs by developing the modular low cost RASCAL concept of liquid hydrocarbon fuel ramjet for tactical missiles.

These two topics will be further detailed hereafter.

1.2 Detonation wave engines technology

Due to its thermodynamic cycle, a detonation wave engine has theoretically a higher performance than a classical propulsion concept using the combustion process. Nevertheless, it still has to be proven that this advantage is not compensated by the difficulties which could be encountered to practically define a real engine and to implement it in an operational flying system.

During past years, MBDA France performed some theoretical and experimental works on Pulsed Detonation Engine (PDE), mainly in cooperation with LCD laboratory at ENSMA Poitiers. These studies aimed at obtaining a preliminary demonstration of the feasibility of the PDE in both rocket and airbreathing modes and at verifying the interest of such a PDE for operational application. Further studies are still in progress with CIAM and Semenov Institute in Moscow. On this basis, several engine concepts have been studied and evaluated at preliminary design level, for both space launcher and missile application. Today, the effort is focused on the development of a small caliber airbreathing engine able to power a UAV with very demanding requirements in terms of thrust range.

The use of a Continuous Detonation Wave Engine (CDWE) can also be considered to reduce the environmental conditions generated by PDE while reducing the importance of initiation issue and simplifying some integration aspects. As it was done for PDE, MBDA France is leading a specific R&T program, including basic studies led with the Lavrentiev Institute of Novosibirsk, to assess some key points for the feasibility of an operational rocket CDWE for space launcher.

But, continuous detonation wave can have also other application for turbojets and for ramjets. In order to address all these possible applications, a ground demonstrator has been designed and should be developed and tested within the next years within the framework of the National Research & Technology Center (CNRT) “Propulsion for Future” located in Orleans/Bourges region.

These studies are also briefly describe hereafter.

2. DUAL-MODE RAMJET TECHNOLOGY

During the two past decades, a lot of system studies, generally based on large technology development efforts, have been performed in France to assess the interest of high-speed airbreathing propulsion for both military and civilian application ([1] to [10]).

The development of such operational, civilian or military, application depends of two key points :

- development of needed technologies for the propulsion system as a low weight, high robustness fuel-cooled structure for the combustor,
- capability to predict with a reasonnable accuracy and to optimise the aero-propulsive balance (or generalized thrust-minus-drag).

2.1 Technology development effort

Even if technologies will finally need to be flight proven, a large part of the technology development effort can be led with available ground test facilities [11] and classical numerical simulation (thermics, mechanics…).

In that field, the effort started during the PREPHA Program has being continued last years through several initiatives taken by ONERA and MBDA France to maintain and develop knowledge and preserve human and material investments in spite of the lack of new National or European R&T program [12] :

- JAPHAR program (ONERA and DLR) ([13] to[18]),
- WRR program (MBDA France and MAI) ([19] to[23]),
- PROMETHEE program (ONERA and MBDA France) ([24] to [27]),

- A3CP (ONERA/SNECMA/Pratt & Whitney)
- PTAH-SOCAR (MBDA France and EADS Space Transportation),
- Other cooperation with research laboratories ([28] to [36]).

Today, the technology development effort is pursued on different aspects which contribute to ensure the performance and thermal and mechanical strength of the combustion chamber :

- variable geometry needed to optimize the performance on the overall flight Mach number range,
- fuel used as coolant for combustion chamber structure (endothermic fuel **-** [37] to [41]),
- fuel-cooled structure itself ([42] to [51]).

In the field of fuel-cooled structures, several C/SiC composite panels have been successfully tested (Fig.1) in representative conditions and long accumulated test duration. This effort led to the development of a part of a combustion chamber duct, made of one single part, which has been successfully tested at ONERA ATD 5 test facility in early 2006 (Fig.2).

Fig.1 – Example of fuel-cooled panel tested in 2004

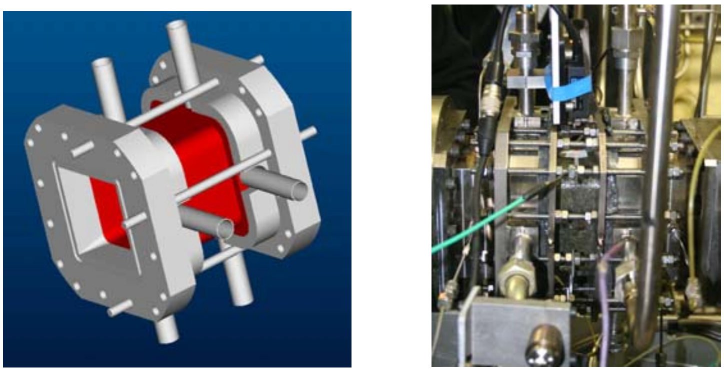

Fig.2 – Cooled composite combustion chamber duct (in red) tested at ONERA ATD5

Regarding the studies related to endothermic fuels, the effort is shared between reforming kinetic modelling (cooperation with DCPR in Nancy) and basic experiment at ONERA to understand the reforming process and to validate the modelling (Fig.3). These works allowed to develop a numerical tool able to simulate the operation of a fuel-cooled structure taking into account the heat exchanges, the fuel hydrodynamics and its reforming kinetic (Fig.4).

Fig. 3 – Basic experiment on endothermic fuel reforming at ONERA

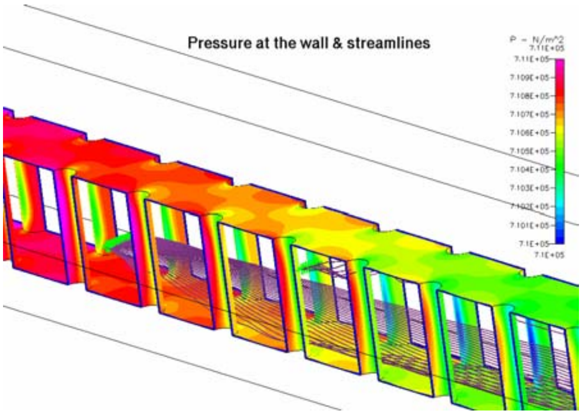

Fig. 4 – Example of coupled simulation of a PTAH-SOCAR fuel-cooled structure

Beyond the works already in progress, the test facility, developed by MBDA France and ROXEL in their Bourges Subdray test center in the framework of PREPHA program (Fig.5 , [49]), is under upgrading. The new test facility, called METHYLE, will allow performing long endurance test in representative conditions to pursue and reinforce technology development by using a modular water-cooled dual mode ramjet combustion chamber able to integrate different kind of testing parts as for :

- element of variable geometry,
- sealing system,
- fuel-cooled structure,
- measurement techniques,
- engine control system…

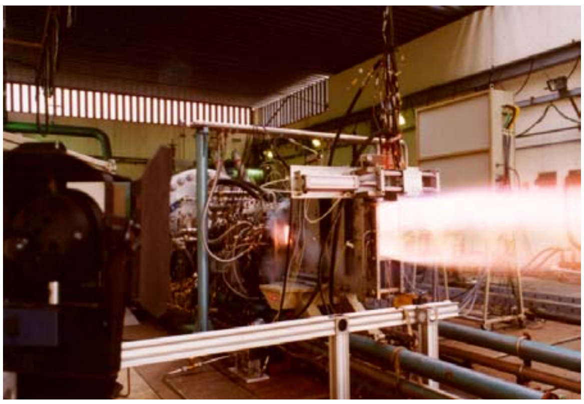

Fig. 5 – Hypersonic test facility at MBDA/ROXEL test center to be ugraded to METHYLE configuration

2.2 LEA flight test program

2.2.1 Aero-propulsive balance sensitivity

For an airbreathing propulsion system, the higher the flight Mach number, the more sensitive the net thrust. At Mach 8, for example, an error of 5 % on nozzle performance leads to a reduction of 35 % in net thrust.

Then, it is more and more mandatory to optimize the integration of the propulsion system into the vehicle airframe and propulsion system components are operating in a very coupled way which would require to test the overhall system to determine the global performance.

But, when the considered flight Mach number increases, it becomes more and more difficult to simulate right flight conditions with on-ground test facilities. Generally, in such test facilities, air is heated up to total temperature before being accelerated through a nozzle to enter the test section at the right Mach number. What ever the heating process may be, that generally leads to the creation of radicals, and very often some pollution, into the feeding air which can change combustion process.

By another way, some scaling effect are difficult (or impossible) to solve with similarity rules. Then, the overall system should be tested at full scale that implies very large, and extremely expensive, if feasible, test facilities.

2.2.2 Development methodology

The extreme sensitivity of the aero-propulsive balance on one hand, and the limited capability of ground test facilities to represent right flight conditions on the other hand make mandatory the definition of a specific on-ground development methodology coupling very closely experimental and numerical approaches. In such a methodology, the in-flight performance can be predicted only by a nose-to-tail numerical simulation. Then on-ground test facilities will be used to performed partial test of vehicle and propulsion system components separated or coupled two by two.

These tests have two goals :

- to allow components design tuning and verify a minimum performance,
- to verify, step by step, the ability of numerical simulation to predict accurately performance in conditions as close as possible to the actual flight.

Obviously, such a methodology is very challenging. So, before starting any operational development, it must be demonstrated that applying this approach will give an accurate value of the performance, allowing to guarantee design margins and to identify properly right directions for optimizing system design. That is why, a flight experimental program is a mandatory step towards future operational developments.

Beyond all current technology development works mentioned hereabove, and on the base of previous acquired results, MBDA France and ONERA started a flight test program, called LEA, in Januar 2003 with the support of French Administration.

In order to limit the cost, this flight test program will be realized with a minimum experimental vehicle without any technology demonstration purpose (use of existing technologies as often as possible) (Fig.6). In the same view this vehicle will be non-recoverable, then non-reusable.

The test principle consists in accelerating the flight experimental vehicle specimen thanks to an air-launched booster up to the given test Mach number, chosen in the range 4 to 8. Then, after booster separation and stabilization, the experimental vehicle will fly autonomously during 20-30 seconds (Fig.7). During this flight, the airbreathing propulsion system will be ignited during approximatively 5 seconds with a fuel-to-air equivalence ratio variation.

The vehicle would be specifically instrumented to give a precise evaluation of the aero-propulsive balance with and without combustion and to determine the contribution of each propulsion system component to this balance. All measured parameters will be transmited to ground by telemetry and recorded with an on-board data recorder which will be recovered after the crash of the vehicle.

The program aims at performing 6 flight tests, planned between 2010 and 2012 for exploring Mach number range 4 to 8.

As explained previously, and beyond a detailed understanding of the aero-propulsive balance constitution, such a flight test program will give the opportunity to define, implement and validate a development methodology applicable to any future operational development.

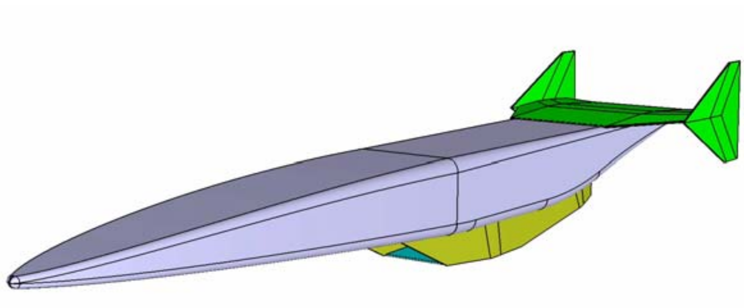

Fig. 6 – CAD view of LEA vehicle

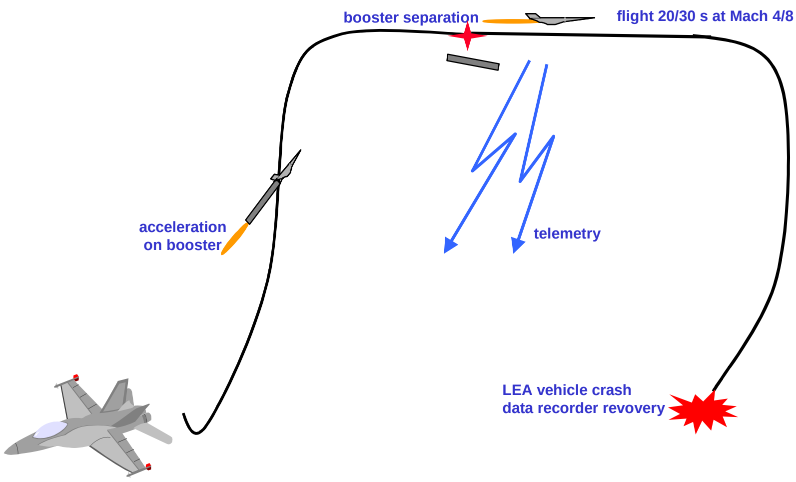

Fig. 7 – LEA flight testing sequence

Since the beginning of 2003, a preliminary design phase has been performed [52].

For the experimental vehicle, the airbreathing propulsion system concept has been chosen by taking into account all results acquired during engines developments performed these last years. The finally selected concept is a variable geometry one using a simple translation movement of the engine cowl and a thermal throttling (Fig.8). Nevertheless, as each flight test will be performed at a quite constant Mach number, a fixed geometry engine will be used on board of each LEA test vehicle, this engine configuration being representative of the selected variable geometry concept at the tested flight Mach number.

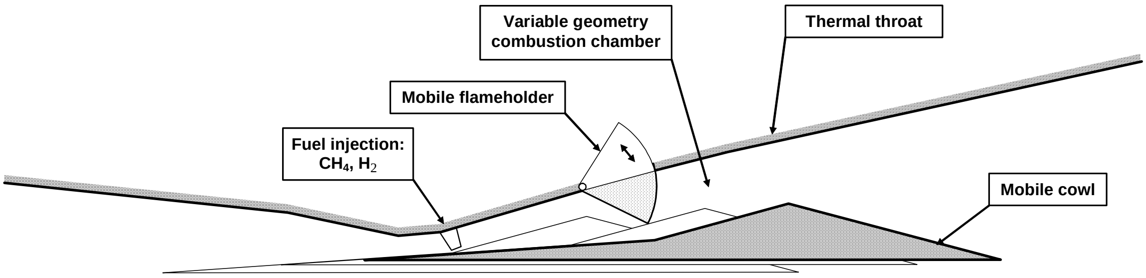

Fig. 8 – variable geometry PIAF engine

A parametric study has been performed on the possible technology for the combustion chamber.The finally selected solution is based on metallic heat sink solution with a high temperature low thermal conductivity coating.

The fuel has also been chosen. The most part of French experience in supersonic combustion is related to Hydrogen. But, considering the very low density of Hydrogen, it is preferable to avoid this fuel in order to limit the size of the tank, then the size of the vehicle and consecutive difficulties to find a possible acceleration system complying with the needs (integration constraints, needed total energy release…).

On the other hand, liquid hydrocarbon fuel could be considered. But, our experience is very limited with such a fuel and it would be difficult to ensure a robust ignition and a good combustion efficiency without previous reforming in a regenerative cooling system (simplest technology used on board of the experimental vehicle).

Finally, a mixture of gaseous Methane and gaseous Hydrogen has been selected. By using this mixture, it is possible to increase the fuel density then limit the fuel tank size. It will be also possible to vary the H2/CH4 ratio during the flight to ensure a robust ignition and control the heat release along the combustor.

Some specific works have been performed to adapt our computation codes to this particular fuel ([53] to [55]). These codes have been validated thanks to basic experiments led in updated ONERA LAERTE test facility. Moreover, ONERA ATD 5 test facility has been updated to allow future CH4/H2 tests for the LEA engine. By waiting, a first test series has been performed with already existing JAPHAR combustion chamber to acquire a first experience with such a fuel [56].

The forebody has been specifically studied. Some parametric studies have been carried out in order to determine a set of design parameters allowing a satisfactory pre-compression while complying with technology constraints.

On the base of an air inlet design and corresponding performances, a first design of the combustion chamber has been realized thanks to 1D, then 2 and 3D computation. On this basis, a full scale mock-up is under manufacturing for future test in ATD 5 ONERA test facility.

Due to the particular configuration of the afterbody/nozzle, a specific effort is still under progress to well understand the interaction between the propulsive jet and the external flow to accurately determine the effect of propulsion on external aerodynamic.

Aerodynamic behaviors of the LEA vehicle and of the Flight Experimental Composite constituted by LEA and its booster have been evaluated by computation for preparing future aerodynamic tests.

Finally, this configuration is under detailed evaluation, including Nose-to-Tail computation, and will be improved step by step.

All the previous elements have been used in a detailed flight simulation in order to obtain a first evaluation of reachable maximum LEA/booster separation conditions. This flight simulation allows simulating a complete flight test sequence including LEA/booster dropping from air carrier, acceleration on booster, separation, descent trajectory of booster, LEA autonomous flight up to final crash.

Other activities have also been carried out to chose the basic technologies used for the LEA vehicle and its propulsion system and a preliminary design has been performed and validated by a Preliminary Design Review.

By another way, a general approach for on-ground testing has been defined but still remain to be refined and confirmed.

Indeed, as Fig.9 shows and on the base of previous studies [57], a large part of the on-ground testing program should be realized in the S4Ma windtunnel located in ONERA Modane test Center in the French Alps.

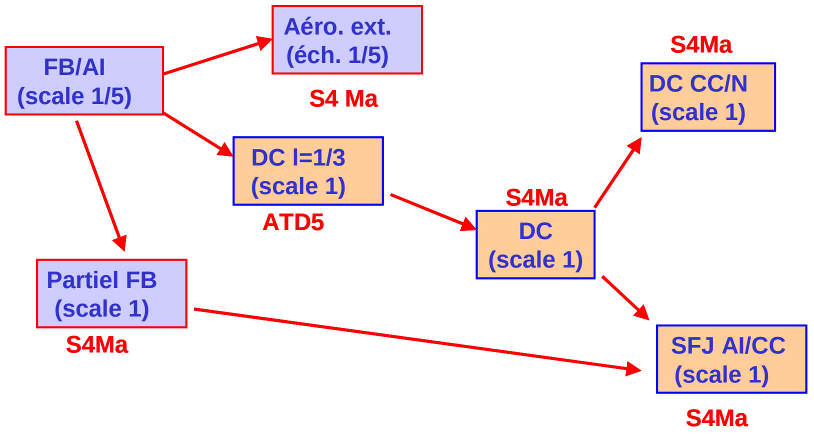

Fig. 9 – general approach for on-ground testing

It is intended to upgrade this test facility in order to take advantage of the existing alumina peeble bed heater which allows to perform test with air non vitiated by water vapor up to Mach 6.5 conditions (1800 K). Thanks to a complementary pre-burner or to an updating of the peeble bed heater, tests corresponding to Mach 7.5/8 flight conditions should be also easily feasible [58].

Detailed design studies, as for example free jet test configuration (Fig.10), have been performed to verify the feasibility of such an upgrading and evaluate precisely the corresponding cost.

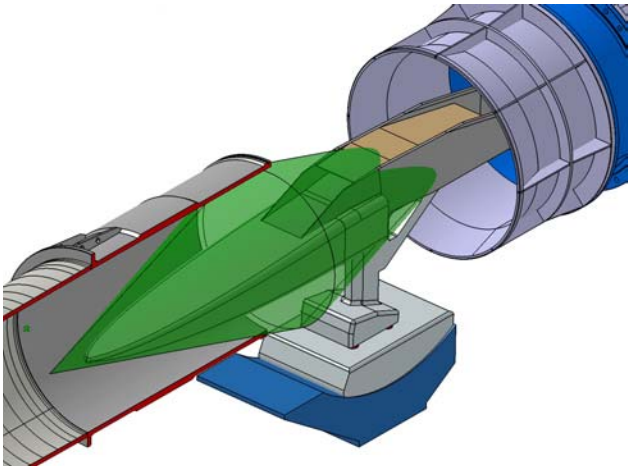

Fig. 10 – Study of LEA free-jet testinstallation in S4Ma ONERA test facility

2.3 Further possible development

The LEA flight test program constitutes a very important first step in the definition and the validation of a development methodology for hypersonic airbreathing vehicles. Nevertheless, if we consider the possible application of high-speed airbreathing propulsion to future reusable space launcher, it is clear that the airbreathing phase will have to be extended up to Mach 10/12 ([4], [7]).

In that view, a minimum R&T program has been proposed [59]. It includes an extension of the flight domain of the LEA vehicle (LEA +) thanks to the upgrading of the present acceleration system or by selecting an other one with higher capabilities. At least, taking into account the corresponding background and associated working partnership, it should be possible to define the most efficient flight test program (in term of scientific and technological return to financial investment).

It has to be noticed that such extended flight experimental program could take advantage of other already existing experimental systems and programs as, for example, the HyShot program which could be used to perform partial technology flight validation for LEA vehicle and propulsion system or for instrumentation.

But, the budget which could be potentially available in Europe within the next years for such a flight test program will be limited. By another way, the on-going LEA flight test program between Mach 4 and Mach 8 has to be first performed. That is why, considering these two points, a proposal has been submitted to ESA regarding a preliminary and less ambitious flight test program, called EAST for European Advanced Scramjet Test, which could be performed by 2010.

The EAST program would consider a subscale (~1/4) twin engines configuration derived from LEA vehicle (Fig.11). EAST would not be a simple supersonic combustion experiment within an academic combustor but would consist in testing the system forebody / air inlet / combustion chamber / partial nozzle during a captive flight on top of a booster derived from a sounding rocket system (Fig.12). The EAST experiment would be fixed on the booster thanks to a strut equipped with a thrust measuring system.

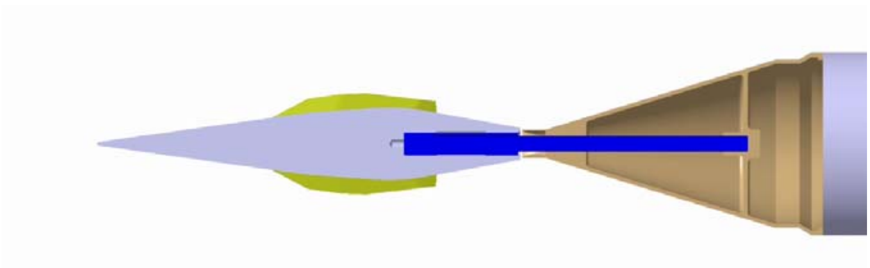

Fig. 11 – EAST configuration

Such a program, dealing with an integrated propulsion system, would allow extending the already defined development methodology by taking into account new ground test possibilities as, for example, high enthalpy short time windtunnels F4 at ONERA Fauga or HEG at DLR Göttingen and to acquire a first flight validation. By another way it would be possible to take advantage of the quite complete propulsion system configuration to flight test the needed improvements of LEA technology to sustain higher flight Mach number conditions.

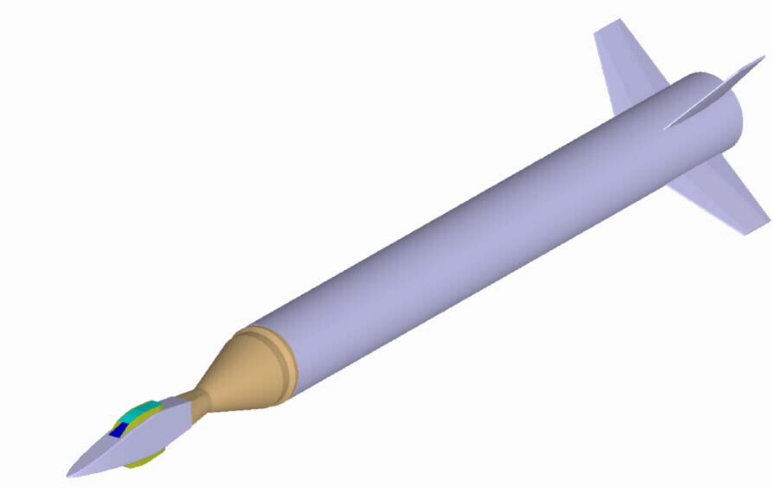

Fig. 12 – EAST on top of the sounding rocket

3. LOW COST LIQUID RAMJET TECHNOLOGY

In France, several studies and advanced developments have been performed on the ducted rocket concept, including choked and unchoked technology [60]. These concepts were initially supposed to provide lower development and production costs than liquid fuel ramjet and considered as more relevant for tactical applications. But, finally, the use of a gas generator for fuel is a real handicap which leads to high development cost and increased development risks :

- The gas generator has to be changed before each combustion test and it is difficult to provide a controlled fuel mass flow evolution during the test facility blowing. As a consequence cost and duration of test series are increased.
- The grain temperature generates variations of the fuel feeding conditions by variation of the grain combustion speed, this parameter has to be explored during the test campaigns. Moreover, it directly impacts the capability to control the global turn-down ratio of the engine.
- When a propulsion system is already existing, the adaptation of the gas generator to different missile requirements needs a new grain development. Mostly the unchocked solution is not adaptable to every mission without defining a grain for each mission, a large flight envelope is not accessible. For the chocked solution, a large flight envelope can be reached but with important unburned fuel losses, due to limited turn-down ratio, limiting the maximum range.
- For boron composition, these losses can be lower than the volumic impulse advantage provided by this fuel, nevertheless the boron fuel gases are very abrasive, very hot for the regulation valve . Boron oxide deposit perturb the operating of the regulation valve .

Considering the limitations of solid fuel ducted rocket, MBDA France chose to develop, with a partial support of French MoD, the low cost RASCAL liquid fuel ramjet concept. The challenge was to dramatically simplify the liquid ramjet technology to obtain very low development and production costs, while maintening a sufficiently high global performance of the ramjet unit and allowing easy and cheap adaptation to different missiles and missions.

The effort was focused on key elements generating cost and mainly the fuel feeding system and the fuel tank pressurisation system.

3.1 Description of the RASCAL technology

The RASCAL ramjet uses a direct injection which consists in injectors mounted on the front dome of the ramjet, a box including the back part of the injectors replacing all the connecting pipes usually needed to feed the injectors located at the end of air inlets diffusors.

To simplify the regulation of the mass flow of fuel injected in the ramjet, on/off injectors are used. The mass flow being regulated by the command of a combination of injectors, no valve is needed for regulation. Several injector families of different individual mass flow are used : the families of more important mass flow permit the access to an important maximum mass flow, the families of smaller mass flow permit the adjustment of the flow with the needed accuracy. Like in a monetary system every value from n g/s to N g/s (where N is the maximum global mass flow) can be delivered by steps of n g/s. With five families of injectors any kind of fuel mass flow law can be provided for any kind of mission and missile thanks to an achievable turn-down ratio up to 20/25. Off-course, depending of the mission requirements (limited altitude range for example), the number of families can be limited to 4 or 3.

The design of injector is very similar for different families of fuel mass flow. To reduce the number of different parts to be manufactured, only one part is different between two different injectors to caliber the mass flow. The adaptation of the ramjet regulation to different ramjet size is very obvious. Two parameters drive this adaptation : the total number of injectors and the mass flow of each family : for small variations of the ramjet size from an existing one, the increase of the number of injectors can be sufficient ; for important difference of size, the change of mass flow families must be considered. In any case resizing injectors is a very easy task and it needs only the resizing of one part.

The regulation valve being suppressed, injectors are used to regulate the mass flow thanks to an electronic card. This very simplified electronic card can be integrated in the engine, for example in the front part of the tank. The on-board computer commands the injector combination trough a very simple algorithm. The mass flow of each combination of injectors under nominal conditions of fuel pressure and fuel temperature are recorded in a table in the computer. During the flight, the control system determines the needed fuel mass flow, then a search in the injectors combination table gives the nearest combination that can be, then, commanded through the electronic card. Obtaining the needed mass flow with a good accuracy is due to the stable fuel conditions delivered by the tank.

The simplification of the tank is mainly based on the simplified pressurization system concept. To avoid the use of complex or expensive systems such as piston driven by hot gases provided by pyrotechnic generators or bladder and turbo pumps, the expulsion power is provided by nitrogen gas stored in a high pressure bottle generally placed inside the tank. The transmission of the pressure to the fuel is done by an elastomeric bladder. A mechanical pressure reducer produces a regulated low level pressure from the high pressure storage bottle. This pressure reducer is maintaining a constant injection pressure drop allowing the injectors operating over steady conditions. The high pressure bottle is a very cheap winded structure, the bladder is a casting of elastomer. The cost of the tank is by that way dramatically decreased.

The extension of the missile range can be easily obtained by extension of the tank length. The qualification of the new tank needs only structural tests. No operating test of the system are needed as for pyrotechnic gas generators. The qualification of the new tank is for that reason very cheap compared to solid fuel solutions.

The assembly of the engine is simplified. Tank is assembled with the injection dome on one hand. The integral booster is cast in the ramjet combustor structure on the other hand. The two part are then finally assembled all together with pyrotechnic igniters. By this way, the costly pyrotechnic phase of assembly is strictly limited to the very ultimate assembly phase.

3.2 Feasibility demonstration of RASCAL concept

The demonstration of the fuel expulsion by an elastomeric bladder and a high pressure gas bottle was first carried out. A high pressure Nitrogen bottle was located in the center of a 225 mm diameter tank. The extensible bladder was placed around the high pressure bottle at the begin of the expulsion and near the tank structure at the end of the fuel expulsion. An expulsion rate of 97% was demonstrated and the model for gas expansion from the bottle validated. It is worth to notice that this result is very good considering the absence of fuel circuit and corresponding un-usable fuel volume.

The development of the on–off direct injection system was then carried out with internal funding by MBDA France from 1993 to 1997. General studies of the injection layout were performed. The injector operating was demonstrated and the electrical spending necessary for a correct opening and closing evaluated. The injected fuel spray was optimized.

Then, tests have been performed to validate that the spray delivered by the on–off injectors was efficient for combustion and mainly that the commutation of injectors configurations didn't perturb the combustion. Tests, performed on an existing configuration of heavy wall mock-up ramjet, validated a ramjet operating with expected performances and no perturbation of the combustion by the injectors commutations.

A new test series allowed to reach the following objectives :

- demonstration of an air-to-air flight envelope capability,
- demonstration of the ramjet robustness against air inlet distortion during maneuvers,
- evaluation of ramjet performances in the air-to-air flight envelope.

The RASCAL injection dome was coupled with a 180 mm caliber ramjet combustion chamber based on a "two 90° air inlets" configuration easily compatible with airplane integration (Fig.13). 12 injectors have been used to cover the needed fuel mass flow range. The operating of the ramjet in the air-to-air complete flight envelop was demonstrated. No important effects of air inlets distortion were observed up to 10° of angle of attack while no combustion instabilities were measured.

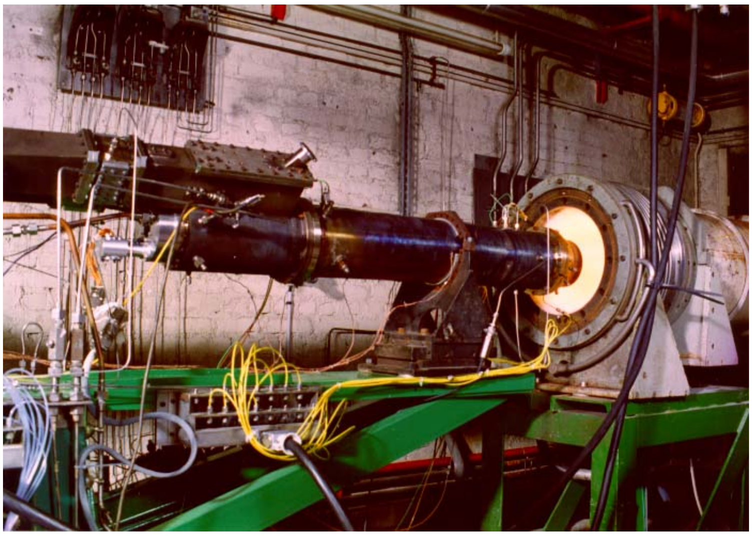

Fig. 13 - RASCAL Combustion test in ONERA Palaiseau test facility

After these works, the French MoD decided to fund complementary tests of the system to demonstrate :

- the capability of the fuel tank to operate in a very large temperature range,
- the operating of the complete engine, the fuel tank and the ramjet being coupled during a complete trajectory simulation (synthesis test).

Expulsion test were performed with fuel tank and injection dome at very low then very high temperature to demonstrate the capability of bladder and of on-off injectors to operate in every operational conditions.

Computer based command control system and fuel regulation algorithm were used for these tests. Fuel expulsion and regulation were successful in every case. The fuel mass flow accuracy on a trajectory profile was demonstrated. The on-off injectors encountered no temperature problem and are so qualified for extremely low and high temperatures.

Two synthesis test of a complete RASCAL system were then performed on the 180 mm caliber configuration. The test configuration included :

- tank with high pressure bottle, bladder and pressure reducer,
- ramjet with combustion chamber, air inlets internal shapes and turning valve for flame stabilization,
- injection dome including 12 injectors,
- computer based command control system and fuel regulation algorithm with equivalence ratio loop.

Two trajectories were selected for these synthesis tests, a low altitude type and a low-high-low type. The test sequence simulated a missile flight with opening of the pyrotechnic closing valve, then injector command for ramjet ignition followed by the trajectory flow variations. The tank that had been at this time only separately tested was connected to the injection dome of the ramjet and feed it during the ramjet combustion.

The two tests were completely successful. The capability of the tank to properly feed the ramjet all along the trajectory was demonstrated. The pressure reducer operated correctly regulating the injection pressure drop.

The capability of the regulation system, including injectors and regulation algorithm, to regulate with a good accuracy the ramjet was demonstrated. The high altitude trajectory culminating at 16 km was perfectly simulated as well as the sea level one :

- no interaction between combustion and tank operating were observed,
- no combustion instabilities appeared during the test,
- the ramjet performances confirmed those obtained during separated ramjet combustion test.

These synthesis tests confirmed all the operating predictions and completed the demonstration that the RASCAL concept covers the flight envelope needed by any kind of tactical missile.

3.3 Further development works

A new study was undertaken, under the aegis of the French MoD, for assessing the performance of the concept for larger engine caliber while demonstrating integral booster compatibility and injectors reliability in operational conditions.

3.3.1 Larger combustor tests

The RASCAL pre-development tests had been performed in 180 mm caliber and with 2 air inlets located beside the combustion chamber at 90° each of the other. This configuration, as described previously, gave very promising results.

To verify the possibility to use RASCAL concept for larger engines, new test series have been performed with a 350 caliber ramjet. These test series mainly aimed at confirming the level of performances measured in small caliber (Fig.14).

A 2 air inlets at 180° configuration has been chosen. A heavy wall mock up has been designed and manufactured. The injection dome integrates 24 injectors of 5 fuel mass flow families (Fig.15). This configuration allows to vary the liquid fuel mass flow from 1 to 60 that is necessary to get access to a sufficient altitude range coupled with an efficient ramjet regulation. For this ground testing, the injectors of different sizes can be moved on the dome. By this way, the injectors positions can be optimized for obtaining the best fueling of the flow. This modularly designed mock-up permited to choose an optimized combustion stabilization and injection system.

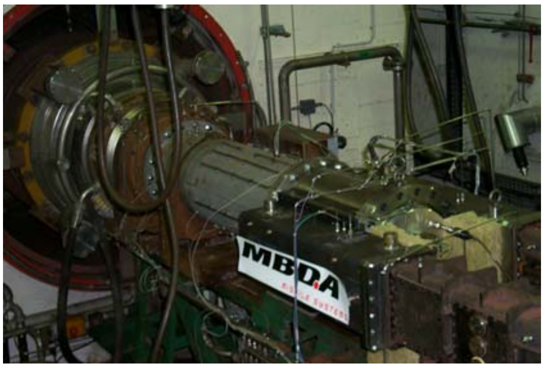

Fig. 14 – RASCAL 350 mm Ramjet model

Fig. 15 – RASCAL 350 mm Injection dome

Combustion test series, performed at ONERA Palaiseau ramjet test facility, allowed assessing the minimum achievable performances in the whole flight envelope and determining operating limits at different altitudes. Combustion efficiency and available thrust were evaluated and some performance improvement possibilities identified.

3.3.2 Injectors reliability

As the engine control is based on the on-off operating of the injectors, the reliability of these injectors is of-course a key point for the RASCAL concept operational application.

A reliability demonstration programme has then been successfully performed. It included in a first step injection tests after air carriage simulation and in a second step long duration injection tests under free flight vibration environnement simulation.

3.3.3 Injectors / Integral booster compatibility

An other key point for the operational use of RASCAL concept is the compatibility of the injectors with an integral booster. As a matter of fact, injectors are directly placed in the front dome of the combustor and they have then to withstand the very high pressure hot gases during the booster phase while opening properly during the transition phase. Up to the booster/ramjet transition phase, injectors are protected by small covers which must sustain the heat flux and be gas proof to avoid destruction of injectors. They must then be ejected during the transition phase by the fuel pressure for the ramjet phase.

A special test device has been developed for that purpose. A small caliber standard grain, placed in a small case, provided hot gases with representative pressure and duration of the actual integral booster. The front end of the small caliber case included three injectors, with a counterpressure provided by water to simulate the fuel pressure of the real configuration. At the end of the gas generator operating, the injectors covers were successfully ejected and the injectors started to inject the water simulating the fuel.

3.4 Possible applications

Different system studies performed last years confirmed the feasibility and interest of using RASCAL concept for a large set of tactical applications.

3.4.2 Air-to-Air missile

The first design study was performed for air-to-air application within the scope of the UK BVRAAM competition (Fig. 16). MBDA France (former AEROSPATIALE Missiles) teamed with Raytheon to provide the complete ramjet propulsion truck based on RASCAL concept. This proposal allowed demonstrating the fully compliance of the ramjet with long range high-g manoeuvering air born missiles : the ramjet providing an important range to weight ratio and the RASCAL technology an economic solution. Moreover, this competition gave the opportunity to perform a complete exercice of quotation for both development and serial production which actually demonstrated the low cost characteristics of RASCAL concepts.

Fig. 16 – Air-to-Air FMRAAM missile / RASCAL configuration

3.4.2 Supersonic target

A supersonic target was also studied. For this, the RASCAL ramjet has been adapted to a 350 mm vehicle able of long range with low altitude trajectories with manoeuvers. The initial acceleration was provided by an external booster up to the starting conditions of the ramjet engine. As the ramjet technology had been demonstrated, the proposed project was very reliable and only a few qualification tests were identified as necessary for a development. The absence of integral booster gives a very simplified solution with no inlet covers, no port covers or injectors covers.

3.4.3 Anti-radiation missile

Possible application of the RASCAL ramjet concept to an air-launched anti-radiation system has also been studied. A 250 mm ramjet with an integral nozzle less booster was designed for that purpose. The caliber was determined by the important range necessary for raid operations added to the important warhead necessary to avoid very accurate, then expensive, seeker. The tank and injectors configuration was then adjusted to the mission. It has to be noticed that such a missile would be very easily adaptable to other Air-to-Ground mission.

3.4.4 Range Counter Force and Deep Strike missiles

More recently, application of RASCAL concept to medium range counter force (medium caliber) or to deep strike (large caliber) missiles has been studied in detail. These studies showed that the RASCAL concept could cope with all the needs of such missions and would provide a very high level of performance in term of block speed (> Mach 4) and range.

4. DETONATION WAVE ENGINES

4.1 Pulsed detonation wave engine

The performance of a Pulsed Detonation Engine (PDE) is mainly driven by the following parameters :

- quality of the pre-mixing,
- value and nature of the needed energy for igniting the detonation,
- duration of the high pressure level (depending of the filling coefficient),
- maximum operating frequency depending of the minimum filling time,

and it is not so simple to optimise each of these parameters to obtain a real improvement of the global efficiency.

Moreover, a PDE a priori generates a severe vibration environment, which can induce much more severe requirement for all on-board vehicle equipments or subsystems.

During past years, MBDA France performed some theoretical and experimental works, mainly in cooperation with LCD laboratory at ENSMA Poitiers. These studies, adressing both rocket and airbreathing systems, allowed first to verify the physics and to understand the key parameters effects then to establish and validate a performance modelling [61].

Further works were performed to understand the effect of a nozzle and performance losses induced by the use of air inlet without any closing valve while other studies were assessing the feasibility of several absorbing systems to be used on thrust wall to reduce, even not suppress, the generated vibration environment. On this basis, a first airbreathing engine was designed and tested at LCD (Fig.17). Further studies are still in progress with CIAM and Semenov Institutes in Moscow.

Many authors are focusing their work on high performance concepts in order to take advantage of the high theoretical performance of the PDE. That generally leads to complex arrangements, needed to optimize each of the main parameters of the engine. Such a propulsion systems correspond with high development and production and maintenance costs.

Considering that :

- well known existing propulsion systems (rockets, turbojets, ramjets) have already high performances and are still being improved,
- for each application, corresponding vehicles have been optimized to take the better advantage of these existing engine concepts,

MBDA France does not aim at obtaining the maximum engine performance but try to take advantage of the specific characteristics of the PDE to simplify engine and vehicle conception.

Several engine concepts have been studied and evaluated at preliminary design level, for both space launcher and missile application [62].

Today, the effort is focused on the development of a small caliber airbreathing engine able to power a UAV with cruising and loitering mission which corresponds with very demanding requirement in term of thrust range (Fig.18). Depending of available budget, such a demonstrator could be flight tested within the next few years.

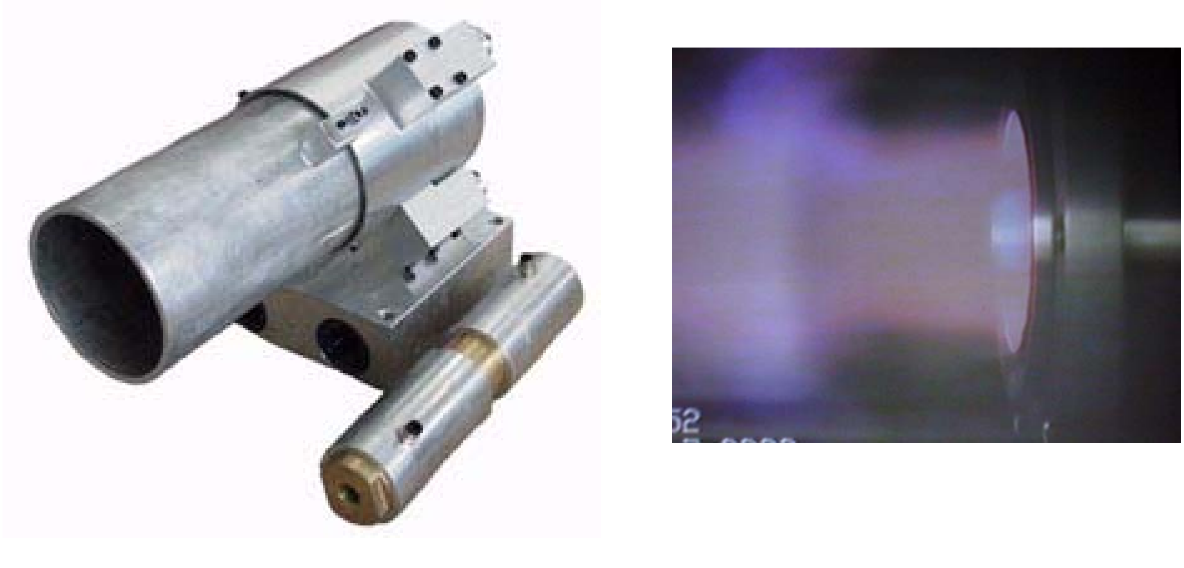

Fig. 17 – First airbreathing engine tested at LCD

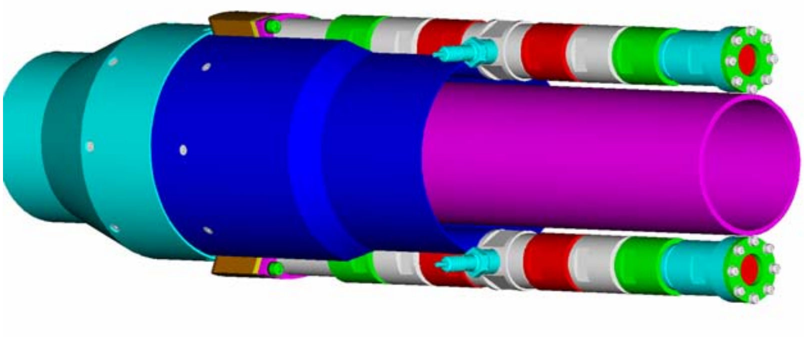

Fig. 18 – UAV demonstrator project to be flight tested within next years

4.2 Continuous detonation wave engine

4.2.1 Principle

The main feature of a CDWE is an annular combustion chamber closed on one side where the fuel/oxydizer mixture injection occurs and opened at the other end (Fig.19). Inside this chamber, one or more detonation waves propagate normally to the direction of injection. In fact a CDWE is very close to a PDE running at very high frequency (typically several kHz) and so the mean pressure inside the chamber is higher than for a typical PDE. If, for a PDE, the injection pressure could be as low as the ambient pressure, in the case of a CDWE the injection pressure should be higher and this kind of engine is better suited for rocket operation than for air-breathing operation.

The flow inside this chamber is very heterogeneous, with a 2D expansion fan behind the leading shock (Fig.20). The transverse detonation wave (BC and B'C') propagates in a small layer of fresh mixture (AB') near the injection wall. The necessary condition for the propagation of a detonation wave is the continuous renewal of the layer of combustible mixture ahead the TDW. The height of this layer *h* must be not less than the critical value *h** for detonation. In the case of a LH_2 / LO_2 engine, the dispersion of liquid oxygen droplets and the quick mixing of the components should be fast enough to decrease the value of *h** and to enable the realisation of CD in small chambers.

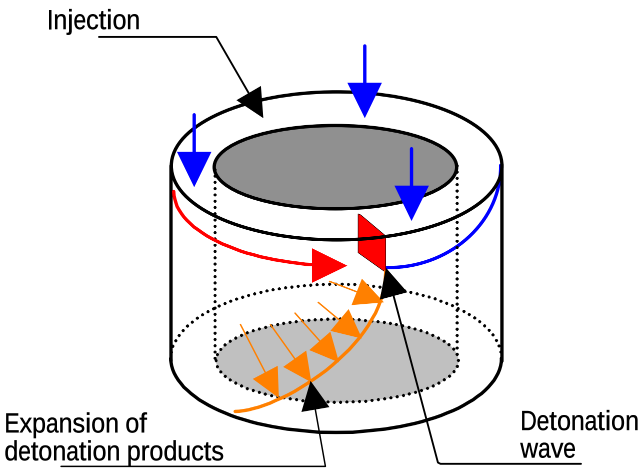

Fig. 19 - Simplified diagram of a CDWE combustion chamber

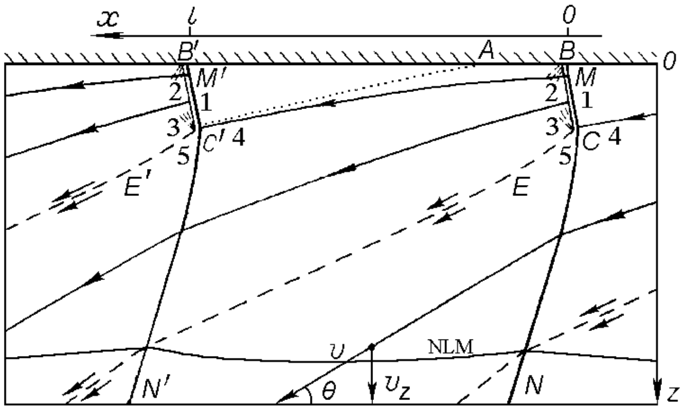

Fig. 20 - Structure of the flow inside the combustion chamber.

The detonation velocity observed in the reference laboratory is generally less than the Chapman – Jouguet detonation velocity. Several explanation could be given to this phenomenon :

- due to the very small time between two detonation waves, the mixing is less than ideal and the detonation characteristics could be changed,
- the fresh gases could partially mix with the detonation products and do not react,
- due to the expansion fan behind a detonation wave and the speed of the flow, the fresh mixture gain some speed in the X direction, so the detonation propagates in counter-flowing fresh mixture.

With this singular flow structure, this kind of engine possess some unique characteristics :

- during the expansion, the flow velocity on the X axis changes direction from –X (just after the detonation wave) to +X (Fig.20),
- the flow velocity on the Z axis could exhibits a transition from subsonic to supersonic inside the constant section duct, through a neutral line of Mach (NLM),
- there is no need for a geometric throat at the end of the combustion chamber,
- the flow exit the combustion chamber at the velocity *v* with an angle θ.

The pressure ratio across the detonation front is typically 10, a value lower than the Chapman – Jouguet detonation wave pressure ratio characteristic (18 for hydrogen – oxygen mixture, and higher than 30 for a hydrocarbon – oxygen mixture).

With the collaboration of the Lavrentiev Institute of Hydrodynamics in Novosibirsk, MBDA-France has been leading some basic experiments. The mixtures used in the different experiments were mainly GH_2 – LO_2 or LHC – GO_2. The goals of those experiments were to address some technical point in order to be able to evaluate the global interest of a engine using TDW for the combustion process. It was found that such engine could deliver impressive thrust in a very small package (275 daN for a 50 mm (internal diameter) and 100 mm long, kerosene – oxygen engine) and that could be increased with the use of a diverging nozzle.

On this basis, MBDA France performed several system studies (including engine design and integration to the vehicle), particularly on rocket engine usable for space launcher [63].

4.2.2 Thrust vectoring investigations

Another peculiarity of a CDWE is that the number of detonation waves inside the chamber is not constant and is a function of the combustible mixture, the combustion chamber geometry and the mass flow rate. As a matter of fact, for a given mixture in a given chamber, changing the mass flow rate (and the injection pressure) will change the number of detonation waves inside the chamber.

As a matter of fact, behind a TDW (and between two consecutive TWD), a complex series of shock waves occurs. If we increase the mass flow rate, the height of the fresh mixture behind two consecutive detonations is sufficient to support a new detonation wave ($h > h^*$) and a shock induced combustion (a detonation) can occur and a new TDW appears.

If we decrease the mass flow rate inside the combustion chamber, the height of the fresh mixture between two consecutive TDW decreases and become not sufficient to support a shock-induced combustion and the TWD degenerates into a more simple shock wave.

Due to the geometry of the combustion chamber, a plug or aerospike nozzle seems to be the best design and the use of self-adaptation of the detonation to the fresh mixture local mass flow made can be optimized to provide a very efficient thrust vectoring with the local increase of the mass flow.

Some experiments were carried out in a 100 mm internal diameter combustion chamber. Fuel and oxidant were injected from the front wall through 190 calibrated holes.

In one series of experiments the equivalence ratio was changed in one half of the engine compared to the other half. In another series, the injectors diameter was increased in order to double the local mass flow rate in one half of the engine.

In all experiments it was possible to obtain an increase of the thrust on one side of the engine. The most promising experiments shows that a 30 % increase of the thrust-wall overpressure was possible if we double the mass flow rate. This increase is lower than expected (ideally a 100 % increase should have been obtained) but the small diameter of the test engine limited the heterogeneity of the flow inside the combustion chamber. With a larger engine, the 100 % increase of the thrust would be possible (Fig. 21).

Compared to a LRE (with a gimbaled nozzle), this thrust vectoring capability is very interesting because it is achievable without any actuator and because the response time is only limited by the response time of the valves controling injection sectors, which could be very fast, enabling the positive control of the vehicle attitude at high frequency without using much power.

The complete pressure field on the injection wall should still be checked, as the effect of the flow deflection and flow expansion on the nozzle on this thrust-vectoring capability.

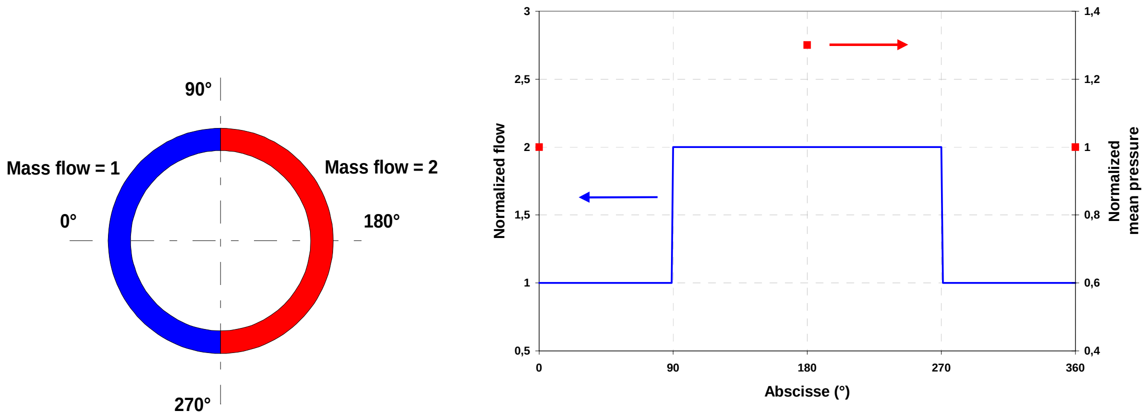

Fig. 21 Thrust vectoring test with mass flow rate control.

4.2.3 Heat fluxes

Due to the transverse velocity of the flow behind the detonation wave (several hundreds of meter per second), the highest heat load inside the CC occurs near the thrust wall and decreases along the axial axis.

This heat flux repartition is very different from the one obtained in a LRE where the maximum heat fluxes occurs near the combustion chamber geometric throat[i]. This point could be beneficial for the engine design because the gasification of the injected oxygen will be faster and the mixing between hydrogen and oxygen will be better.

Of-course, it is then necessary to design an engine structure able to sustain these heat fluxes while the highly fluctuating pressure imposes very specific mechanical loading. In that field, the current step of the work is focusing of the test of engine parts made with composite (C/SiC) materials, sometimes using fuel-cooled structures obtained from PTAH-SOCAR technology (see § 2). The center body is a good candidate for this test that are to be performed with and without active cooling.

4.2.4 Engine components

Based on previous investigations, a actual-size demonstrator has been designed (Fig.22) to be tested in MBDA/ROXEL Bourges-Subdray Test Center. This demonstrator is modular, actively cooled, and will be able to operate with GH_2 / GO_2 or GH_2 / LO_2 with the change of supply lines and injections wall. Even if this test-bench is a non-flying hardware, its modularity will be used to test various engine parts and real components.
Moreover, as continuous detonation wave can have also other application for turbojets and for ramjets, the demonstrator will be able to use air as oxydizer and liquid hydrocarbon fuel.

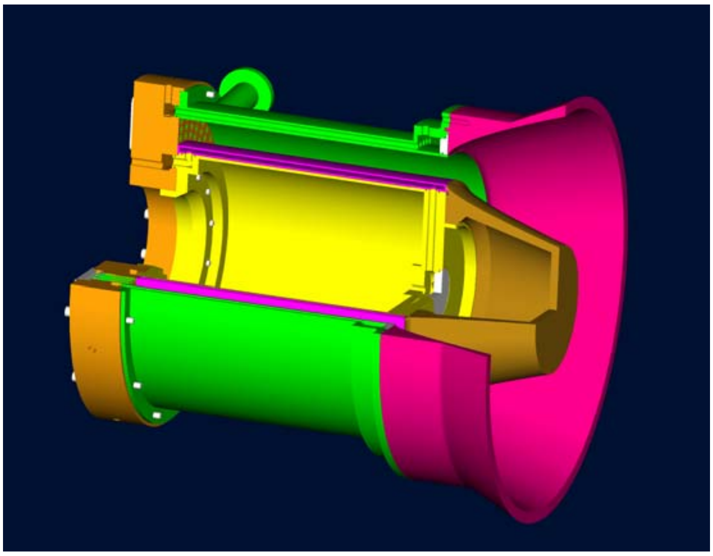

Fig. 22 - cutout of the CDWE demonstrator

The combustion chamber is 350 mm (external inner diameter) and 280 mm (internal inner diameter).

The injection wall is divided in 8 sectors in order to be able to change the local mass flow rate and investigate the thrust vectoring effect with a diverging nozzle or with a center core nozzle (aerospike).

This engine will be used to investigate the flexibility of the CDWE concept and it's capability to operate smoothly in a wide range of injection conditions (pressure, equivalence ratio) in short duration (several seconds) test, but also to check materials resistance in long-duration (several minutes) test.

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